MANEUVER AND TRAJECTORY DESIGN FOR THE ST5 CONSTELLATION

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ABSTRACT

This paper outlines the trajectory and maneuver design of the Space Technology 5 (ST5) mission constellation. Design challenges for the release and deployment of the three ST5 spacecraft into a highly elliptic orbit include collision avoidance, limited delta-v budget, and coupled attitude and orbit maneuver dynamics. The derived requirements levied on ST5's subsystems and the launch vehicle for the successful release and deployment of the constellation are outlined. A maneuver strategy for deployment is given, as well as a delta-v budget showing appropriate margin for contingency.

INTRODUCTION

Space Technology 5 (ST5) is a space technology development mission in the New Millennium Program (NMP) and the National Aeronautics and Space Administration's (NASA) first experiment in the design of miniaturized satellite constellations. Managed by NASA's Goddard Space Flight Center (GSFC) in Greenbelt, Maryland, the mission is scheduled for launch in 2004. ST5 will validate six new space miniaturization technologies, including a Cold Gas Micro Thruster (CGMT) manufactured by Marotta Scientific Controls, Inc. In order to reduce cost, ST5 was designed to be a secondary payload aboard any launch vehicle. The current mission design calls for three 25 kg spin-stabilized spacecraft to be injected from a common launch vehicle into an orbit similar to a Geosynchronous Transfer Orbit (GTO), to achieve a relative formation at apogee. The nominal mission duration is 3 months.

In addition to validating new spacecraft technology, ST5 is also designed for the validation of science technology. Positioned as a pathfinder mission for future missions such as the Magnetospheric Constellation Mission, ST5's science objectives involve validation of the spacecraft platform for obtaining science measurements, determining the spacecraft magnetic field properties, measuring the natural magnetic background, and demonstrating that magnetic phenomena can be measured in-situ (ref. 1). The use of three spacecraft simultaneously taking science observations allows for tracking of magnetospheric phenomena across time and space. Targeted science events include the dipolarization of the magnetic field due to substorm events, magnetospheric compression due to solar wind dynamic pressure, and ion cyclotron waves (ref. 1).

The ST5 science team has recommended observation of these magnetic phenomena across different magnetic regions simultaneously. The desired magnetic separation corresponds to 1.0 hour of Mean Local Time (MLT) between spacecraft at roughly the same distance from the Earth. The definition of MLT in this analysis is the local time of the spacecraft's sub-satellite point, which is the Coordinated Universal Time of the Spacecraft's epoch adjusted by the fraction of a day of the spacecraft's longitude.

The project established a requirement to target 0.5 hour MLT separation at 45 days into the mission, with the eventual goal of achieving 1.0 hour MLT separation passively within 90 days. The 1.0 hour separation provides more time between events between two spacecraft, which is desirable for providing more relaxed orbit determination requirements. This arrangement places less demand on ground based operations and allows for more science telemetry. Therefore, the separation should be made as large as possible.

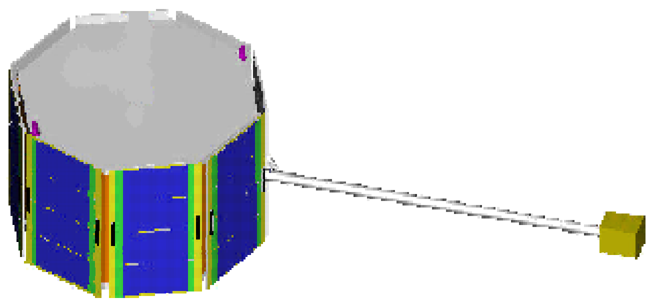

Figure 1: ST5 Spacecraft

Each ST5 spacecraft has a propulsion system onboard which will provide the impulse necessary for sun acquisition, attitude control, and delta-v maneuvers. Shown in Figure 1, each spacecraft is octagonal in shape, with a height of 10.5 inches and side-to-side width of 18 inches. Mounted on the lower deck of the spacecraft, the propulsion tank has a volume of only 145 cubic inches. The CGMT, one of the technologies slated for validation aboard ST5, is required to have a specific impulse of 60 seconds using gaseous nitrogen. Only one CGMT is present aboard each spacecraft. The orientation of the thruster, mounted under the lower deck, allows for pulsed firing over several spin rotations to achieve axial thrust, or spacecraft spin axis precession, or any combination of the two while remaining spin positive. With the tank pressurized at beginning-of-life at 2000 psi, the propulsion system has a design total delta-v capability of 7.6 m/s. Sun Acquisition, Spin Axis Precession Maneuvers, and Spin Axis Pointing Maintenance

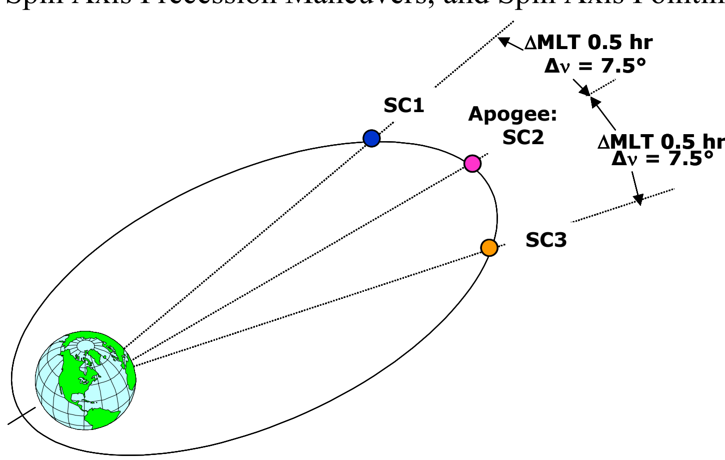

Figure 2: Target Constellation Formation at Apogee

account for 5.6 m/s of the delta-v budget. Therefore the maneuver budget for deploying the spacecraft formation was kept to a conservative 1.0 m/s to allow 1.0 m/s of contingency capability.

The delta-v maneuver was incorporated in the design of the mission as a means of deploying the three spacecraft from the post-release configuration to an in-situ science configuration. While the sun acquisition and spin axis precession maneuvers operate briefly in a 50 msec pulse mode, the delta-v maneuver will operate the CGMT in a 2 Hz 0-to-70% duty cycle mode over several minutes. Figure 2 shows the desired configuration after a successful deployment sequence.

To date a launch vehicle has not been selected to deliver the ST5 constellation. Recognizing that orbit possibilities are limited by the secondary payload status of the mission, a survey of historical data for secondary payload missions was conducted by the ST5 project. This investigation showed that secondary payloads commonly achieve GTO (ref. 2). The highly elliptical characteristics of a typical GTO allow for magnetospheric science capture at apogee, where the relative velocities of the spacecraft are slow, allowing for data rich scenarios. Instead of a final launch profile, the ST5 project has devised a "strawman" profile that is based on historical data of secondary payload orbital parameters. The details of this profile are contained in the ST5 Launch Assumptions Document (ref. 2).

This profile represents a possible initial orbit for the mission and the corresponding timeline currently accepted by the project for the ST5 mission is outlined in Table 1. The launch vehicle upper stage engine cut-off occurs at 1650 seconds after launch. The upper stage is then injected into a 240 kilometer x 37,000 kilometer altitude orbit, with perigee located at the descending node and out of earth's shadow (ref. 1). The first spacecraft released from the launch vehicle, SC1, separates 3700 seconds from launch. The remaining spacecraft, SC2 and SC3, are released at 10 minutes intervals, respectively. After a spacecraft checkout period of about 7 days, the deployment of the formation is executed. This deployment is executed via a series of orbit phasing maneuvers, dubbed "delta-v" maneuvers.

METHOD

The nominal deployment scenario consists of two maneuvers, one executed by each of SC1 and SC3 on days 7 and 8, respectively. These phasing maneuvers are aligned along the velocity direction of the spacecraft in order to take advantage of the largest change in semi-major axis per change in velocity. This approach causes the spacecraft to separate relative to each other at apogee to achieve the MLT separation requirement. These delta-v maneuvers employ a relatively larger amount of sustained thrust than the various pointing maneuvers. SC2, the second spacecraft deployed, has the ability to perform a maneuver on day 9 if necessitated to avoid recontact with the other two spacecraft, but it is not nominally slated to perform one.

The launch vehicle stage from which the ST5 spacecraft are released is assumed to have a battery life after injection of less than one hour. Therefore, the actual launch vehicle separation sequence will not differ much from the current baseline. In contrast, the maneuver plan that will deliver the constellation into its designed formation is flexible and can accommodate off-nominal conditions. In order to determine launch vehicle requirements, the sensitivity of the maneuver plan to deployment errors is examined here. The release sequence errors for consideration are release velocity, attitude, and injected orbit state.

Table 1: ST5 LV Injection, Release, and Maneuver Timeline

Time (L+ seconds)	Event Description	Comments
L+ 0	Launch	
L+ 1650	Upper Stage shut down	Injection. Perigee at descending node
L+ 3700	SC1 Release from LV	Altitude = 1.14 R_E, V = 6.51 km/sec
L+ 4300	SC2 Release from LV	Altitude = 1.53 R_E, V = 5.79 km/sec
L+ 4900	SC3 Release from LV	Altitude = 1.89 R_E, V = 5.22 km/sec
L+ 7 days	SC1 Delta V Maneuver	1.0 m/s velocity direction
L + 8 days	SC3 Delta V Maneuver	1.0 m/s anti-velocity direction
L + 9 days	SC2 Delta V Maneuver	If needed, mean orbit period of SC1, SC3

The release mechanism is a spring preloaded to provide 1.0 m/s separation velocity between the launch vehicle and the spacecraft. Similar to the delta-v maneuver, the direction of the release is required to be along the velocity vector to maximize separation from the launch vehicle. While it is standard for major launch vehicle deliveries to perform a Contamination and Collision Avoidance Maneuver (CCAM), no such assumption was made in planning the ST5 release. For the purposes of this analysis, the release mechanism will be assumed to operate nominally. Any further mention of velocity error will refer to off-nominal alignment of the launch vehicle at release due to attitude error. Finally, the injected orbital state error, within the 3σ tolerance presented in Table 2, has a negligible effect on the problem since only the relative dynamics of the system are of interest.

Table 2 : ST5 Injection Orbital State

Element	Value	Tolerance (3σ)
Apogee (km)	37,000	$\pm$ 93
Perigee (km)	240	$\pm$ 5.6
Inclination (Degrees)	20.5	$\pm$ 0.03
Argument of Perigee (Degrees)	180	$\pm$ 2

Any off-nominal pointing of the LV during the release sequence will deliver less delta-v in the desired direction than expected. The immediate post-release dynamics are affected by the magnitude of this alignment error, $\delta\theta_{LV}$. An analysis of the effect of off-nominal alignment of the release velocity on spacecraft-LV dynamics and inter-spacecraft dynamics is presented below. First, the effect of pointing error of the LV on spacecraft-LV relative motion is shown in Figure 3. A preliminary estimate of the orbit determination knowledge at GTO insertion (30 minutes after launch) is 40 km in-track.

Therefore, a 40 km proximity constraint was imposed upon the closest approach of the spacecraft to the vehicle after release. From the plot, the value of the alignment error that drives the closest approach to the launch vehicle to 40 km is 62°.

Next, the $\delta\theta_{LV}$ allowable to prevent the collision of ST5 spacecraft with each other are determined from review of the release sequence. A large enough misalignment for SC1's release combined with a nominal release for SC2 will result in both spacecraft having very nearly the same orbit period, thus maintaining short term proximity within orbit knowledge boundaries. This condition is avoided by defining the launch vehicle pointing error requirement further. Figure 4 shows the relative motion dynamics between SC1 and SC2 over the 6 hours following release from the launch vehicle. The origin shown represents SC1. From Figure 4, the value of $\delta\theta_{LV}$ that results in an inter-spacecraft nearest approach of 10 km following release is 10°. Thus, a conservative error bound would be limiting alignment of the release velocity of the lead spacecraft not to exceed 10°.

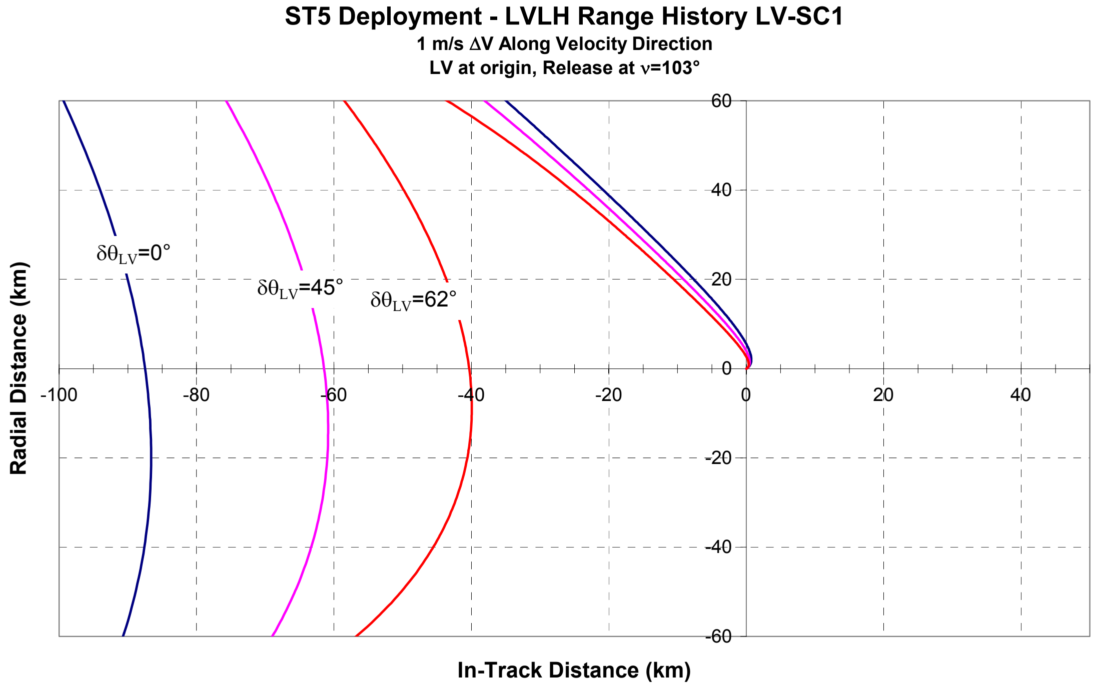

Figure 3: Effect of $\delta\theta_{LV}$ on LV-to-Spacecraft Dynamics at LV Release

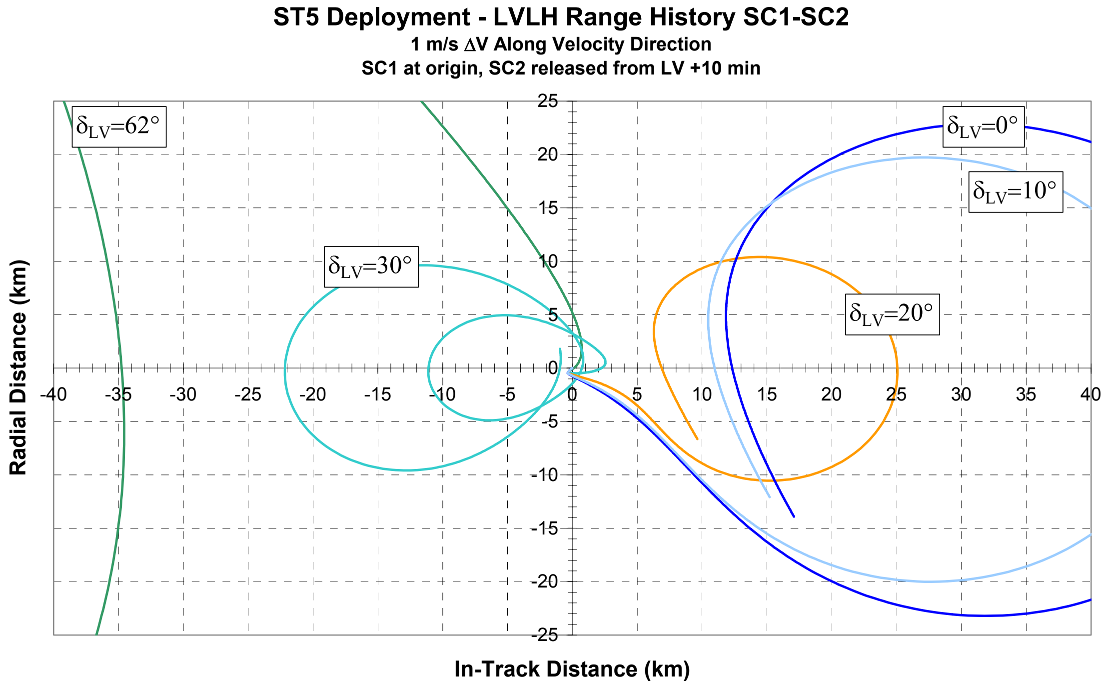

Figure 4: Effect of $\delta\theta_{LV}$ on Inter-Spacecraft Dynamics at LV Release

While some analysis explored possible out-of-plane formations, a "string of pearls" formation, where all spacecraft lie in the same orbital plane, was chosen. Other formations investigated included tetrahedron concepts of up to 6 spacecraft. However, the necessary out of plane components were quickly deemed unobtainable with the on-board propulsion system. Also, MLT goals were not efficiently realized through differential inclination. Given the limited on board propulsive resources, the "string of pearls" is the least complex and most efficient form to achieve. Furthermore, as a launch vehicle selection for ST5 continues and the launch parameters remain unknown, designing a constellation scheme robust to unknown orbital parameters and compliant to the spacecraft operating constraints is crucial for mission success. The constellation eventually became 3 spacecraft as the design was scaled back to meet cost considerations.

The maneuver plan can accommodate faster deployment times using increased delta-v or varying maneuver locations along the orbit. By phasing, exploiting period differences between any two orbits, the ST5 constellation can be deployed into the desired configuration within the desired timeframe. A period difference of 36.7 seconds between any two spacecraft is required to achieve a MLT separation of 0.5 hours in 45 days. Figure 5 illustrates the effect of a 1.0 m/s delta-v maneuver along the velocity direction on the difference in period (ΔP) from the ST5 nominal orbit as a function of true anomaly. The solid line models the small impulse relation to delta-v:

$$\frac{\Delta P}{\Delta V} = 6\pi\sqrt{\frac{a^5}{\mu^3}}V$$

where a is the semi-major axis, μ is the gravitational constant of the Earth (398,600.5 km^3/sec^2), and V is the instantaneous orbital velocity.

The maneuver is modeled impulsively for this graph. From this relation, the orbit position in true anomaly suitable for the delta-v maneuver can be extracted. Notable is the inability to achieve the formation goal in 45 days when the maneuver is performed near apogee. An envelope of ± 55° about apogee is the exclusion zone where a delta-v maneuver will fail to achieve the 0.5 MLT separation in 45 days. Also notable from the above equation and this graph is the dependence on semi-major axis, currently 24,998 km. The impact of a final orbit injection semi-major axis larger than 24,998 km is an exponential growth in period difference achievable by the 1.0 m/s maneuver. Larger

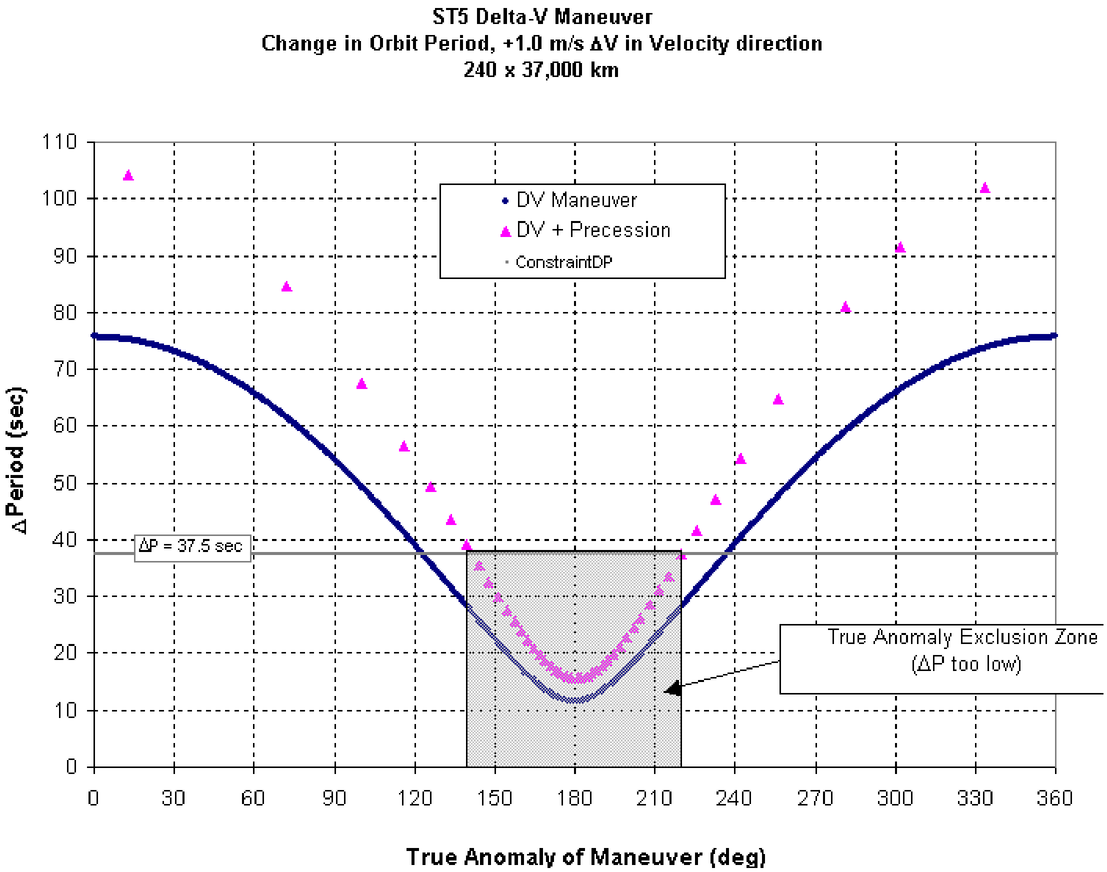

Figure 5: Change in Orbit Period vs. Orbit Position of Maneuver Showing Minimum ΔP Required and Maneuver Exclusion Zone

final orbits are better for achieving the maneuver target, as they make the exclusion zone smaller and provide higher ΔP for a given orbit position, thus higher separation rates.

When the delta-v maneuver is modeled with the necessary pre- and post- attitude maneuvers, this envelope becomes reduced. ST5's spin axis is nominally co-aligned with the ecliptic north axis. Prior to the delta-v maneuver, the spin axis will have to be rotated to become co-aligned with the spacecraft velocity vector at the midpoint of the maneuver. In performing this attitude maneuver, a component of the delta-v used to rotate the spin axis also combines with the nominal delta-v maneuver along the velocity vector. A simple analytic model of this "additional" delta-v, which assumes small finite burn times for the attitude maneuver and an orbital inclination of 20.5°, produces the results in Figure 5. The exclusion zone envelope is reduced by the contribution of the attitude precession maneuvers to ± 40°. While the results here are dependant on the assumed relationship between the orbit plane and the ecliptic plane, which will depend on the final launch parameters, the qualitative trend is such that the attitude maneuvers pre and post delta-v will increase the inter-spacecraft separation rate produced by the nominal delta-v.

The preferred position for maneuver planning is the right side, or exit side, of the exclusion zone. It is anticipated that any timing errors or delays in the execution of the maneuver would serve to increase the separation rate of any two spacecraft. A maneuver on the left side of the exclusion zone, or entrance side, would risk too small of a maneuver to achieve the mission formation required. There is no maximum separation rate acheivable by the 1.0 m/s maneuver that is undesirable to the science team, therefore risk is minimized by manuevering on the exit side.

Within the acceptable orbit envelope for the delta-v maneuver of ± 140° centered about perigee, the attitude control system and propulsion system must perform within certain error bounds. The performance of each system is deemed independent of each other and thus will be derived from an overall error via a root sum square relation. First, a minimum inter-spacecraft separation rate will be derived from the maximum allowable parameters that define a successful formation deployment. Next, this off-nominal separation rate will be resolved into an error parameter for delivered delta-v of the maneuver. Finally, the error will be equally allocated to the Attitude Control System (ACS) and propulsion system in terms of pointing requirement and a percentage performance requirement, respectively.

Maneuver accuracy requirements will drive the propulsion system performance requirement. The target time to achieve this goal, T, is 45 days, or half the mission lifetime. The nominal drift rate at apogee, is

$$\Delta M\dot{L}T_{NOM} = \frac{\Delta MLT_{goal}}{T} = \frac{0.5hr}{45day} = 0.011\frac{hr}{day} = 0.17\frac{\deg}{day}$$

The time error allowable to achieve the formation, δT, will be assumed to be 15 days. This allowable time was set by the ST5 project as a bound on the target for deployment. It is one half of the time remaining in the mission from the initial target of 45 days. Therefore, the minimum separation rate, is

$$\Delta M\dot{L}T_{MIN} = \frac{\Delta MLT_{goal}}{T + \delta T} = \frac{0.5hr}{(45+15)day} = 0.0083\frac{hr}{day} = 0.13\frac{\deg}{day}$$

Converting these angular rates to separation rates in km/orbit using the half angle formula with the distance from the earth's center at apogee, R_A, and orbit period, P:

$$\dot{\Delta s} = \frac{P}{24} 2R_A \sin\left(\frac{\Delta \dot{MLT}_{NOM}}{2} \cdot 1day\right) \frac{km}{orbit},$$

$$\dot{\Delta s} = 57.3 \frac{km}{orbit},$$

$$\therefore \dot{\Delta s}_{MIN} = 43.0 \frac{km}{orbit}$$

P = 10.9 hr/orbit; R_A = 43,378.14 km

The orbit period difference required to achieve this; ΔP; can be approximated from the separation rate, expressed in km per orbit, and the velocity of the spacecraft in its initial orbit at apogee, V_{apogee}, expressed in km per second:

$$\Delta P = \frac{\dot{\Delta s}}{V_{apogee}} = \frac{57.3 \frac{km}{orbit}}{1.56 \frac{km}{\sec}} = 36.7 \frac{\sec}{orbit},$$

ΔP_{MIN} = 27.6 sec/orbit

ΔV is proportional to ΔP for small impulses, therefore the delivered ΔV requirement expressed as a relative percentage is

$$\frac{\Delta V_{MIN}}{\Delta V_{NOM}} = \frac{\Delta P_{MIN}}{\Delta P} = \frac{27.6}{36.7} = 0.752 = 75.2\%$$

Therefore the delivered delta-v relative error allowable, $\varepsilon_{\Delta V}$, is

$$\varepsilon_{\Delta V} \equiv \frac{\delta \Delta V}{|\Delta V|}$$

$$\varepsilon_{\Delta V} = \frac{\Delta V_{NOM} - \Delta V_{MIN}}{|\Delta V_{NOM}|}$$

$\varepsilon_{\Delta V}$ = 1 - 0.752 = 0.248 = 25%

The relative error, $\varepsilon_{\Delta V}$, consists of two independent error sources: the propulsion system performance error, δ_{PROP}, and the attitude control system performance error, δ_{ACS}. In order to derive δ_{PROP} and δ_{ACS}, these quantities are weighted equally thus, set equivalent in relative form, ε.

$$\varepsilon_{ACS} = \varepsilon_{PROP}$$

Combining the pointing relative uncertainty for the ACS, ε_{ACS}, and the propulsion system performance relative uncertainty, $\varepsilon_{prop,}$ via root sum square yields:

$$\sqrt{\varepsilon_{ACS}{}^{2} + \varepsilon_{PROP}{}^{2}} = \varepsilon_{\Delta V} = 0.248,$$

$$\therefore \varepsilon_{ACS} = \varepsilon_{PROP} = 0.175 = 18\%$$

Converting to absolute uncertainty values:

$$\varepsilon_{PROP} \equiv \frac{\delta_{PROP}}{|\Delta V_{NOM}|}$$

$$\delta_{PROP} = \Delta V_{NOM} (\varepsilon_{PROP})$$

$$\boxed{\delta_{PROP} = 0.18 \text{ m/s}}$$

Noting that an alignment angle error, $\delta\theta$, relates to δ_{ACS}:

$$\varepsilon_{ACS} \equiv \frac{\delta_{ACS}}{|\Delta V_{NOM}|}$$

$$\delta_{ACS} = 0.18 \text{ m/s}$$

$$\delta\theta = \arccos\left(\frac{\Delta V_{NOM} - \delta_{ACS}}{\Delta V_{NOM}}\right)$$

$$\delta\theta = \arccos(1 - \varepsilon_{ACS})$$

$$\boxed{\delta\theta = 35^{\circ}}$$

This results in a pointing error requirement levied on the ACS for the delta-v maneuver not to exceed 35° and a propulsion system performance minimum requirement of 82% of nominal.

CONCLUSION

The ST5 maneuver plan incorporates a method of deploying three spacecraft into an expansive formation at apogee of an elliptical orbit. A "string-of-pearls" formation is simple to implement and allows for large scenario changes from the current one with regards to initial orbit, deployment time, and delta-v required. Out-of-plane formations involving plane changes and/or inclination changes were too expensive for the propulsion resources aboard ST5 spacecraft and the given scenario.

In order to maximize the separation of spacecraft from each other and the launch vehicle at release within the given time frame, the pointing requirement on the launch vehicle is no more than 10 degrees off the velocity direction at release. While the CCAM maneuvers most likely performed by the launch vehicle were not included in this analysis, the envelope of worst case conditions were developed for use in future constellation mission analysis.

The derived requirements of ST5 systems in support of the delta-v maneuvers are summarized. The orbit position envelope allowing for successful deployment of the ST5 spacecraft using a 1.0 m/s delta-v maneuver is ± 140° in true anomaly centered about perigee, which represents a 3.2 hour arc of a 10.2 hour orbit. The pointing requirement levied on the ACS in support of delta-v maneuvers is not to exceed 35° off nominal. The CGMT performance must be predictable to no less than 82% of the nominal maneuver delta-v.

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